Thermal Control Architecture Trade Study for the Europa Clipper Pre-Project Study

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This paper details the technical trade study conducted for the Europa Clipper spacecraft concept between active and passive thermal control architectures: specifically on the methodology, analysis, and results used to determine the thermal subsystem power demands for both architectures. The trade study results demonstrate the active thermal control architecture would be significantly more energy efficient during the science observation phase of the mission. The Trade study also brought to light some intricacies of the thermal architecture that will be further investigated in the future to improve the design and performance of the baseline thermal subsystem. The Europa Clipper Pre-Project Study is a mission concept under investigation in a partnership between the Jet Propulsion Laboratory and the Applied Physics Laboratory.

Nomenclature

AFT = Allowable Flight Temperature
APL = Applied Physics Laboratory
A.U. = Astronomical Unit
BV = Bypass Valve
CAD = Computer Automated Design
CBE = Current Best Estimate
CFC-11 = Trichlorofluoromethane (Freon)
HGA = High Gain Antenna
IMU = Inertial Measurement Unit
HRS = Heat Rejection and Recovery System
IPA = Integrated Pump Assembly
JPL = Jet Propulsion Laboratory
ME = Main Engine
MLI = Multi-layer Insulation
MMOD = Micrometeoroid and Orbital Debris
MSL = Mars Science Laboratory
PCA = Pressurant Control Assembly
PCU = Power Conditioning Unit
PIA = Propellant Isolation Assembly
RF = Radio Frequency
RTG = Radioisotope Thermoelectric Generator
SA = Solar Array
SRU = Stellar Reference Unit
TWTA = Traveling Wave Tube Amplifier
WU = Reaction Wheel Unit

I. Introduction

As a response to the 2011 Decadal Survey [1], NASA's Jet Propulsion Laboratory (JPL) and John Hopkins University's Applied Physics Laboratory (APL) developed the Europa Clipper Mission Concept. Europa is one of the leading contenders for celestial bodies in our solar systems that may harbor life. As such, the mission concept would use a variety of science instruments to characterize the ice shells, subsurface ocean, surface topography, and magnetic environment of the icy moon with the goal to improve our understanding on habitability of Europa and other similar icy moons. Previous Europa mission concepts have had a difficult time tackling the harsh radiation environment near and around Europa. To address this problem, the Europa Clipper mission concept employs a unique trajectory: a long looping orbit around Jupiter with multiple flybys of Europa. During the flybys, the instrument payload obtains science and reconnaissance data of the icy moon. After each of the flybys while in an environment with lower radiation levels, the spacecraft relays the data to earth and is also able to recharge the batteries in preparation for the next flyby. Additionally, the majority of the spacecraft and instrument electronics would be housed inside a mechanical vault structure that provides further protection from the local radiation environment at Jupiter.

The reference spacecraft employs two wings of solar arrays with $53m^2$ of active area as its power source, (Figure 1). But as an outer planet mission, power production and consumption are limited. Additionally, thermal power demands tend to be one of the largest out of all the subsystems for outer planet missions. Previous outer planet missions of similar sized spacecraft have employed passive thermal control architectures: NASA's Juno, Cassini, and Galileo. Up until very recently, these spacecraft have always relied on the use of Radioisotope Thermoelectric Generators (RTGs) and other nuclear sources for electrical and thermal power. Juno, slated to arrive at Jupiter in 2016, will be the first Jupiter mission to use Solar Arrays. And while this spacecraft uses a passive thermal architecture of louvers, doublers, and electrical heaters, the mission did not employ a Venus flyby in its trajectory. Europa Clipper's unique combination of a solar powered outer planet mission with a Venus flyby trajectory, along with the success of mechanically pumped fluid loops used in recent interplanetary missions, have provided enough need and merit for the mission to consider the use of an active thermal architecture employing a Heat Rejection and Harvesting System (HRS). The main impetus behind this consideration is the fluid loop's (HRS's) ability to very efficiently harvest waste heat from the power dissipating components (e.g., avionics, telecom, etc.) for use in the non-dissipating subsystems, like the propulsion module, leading to a significantly reduced power/energy demand for thermal control, when compared to a passive system [2].

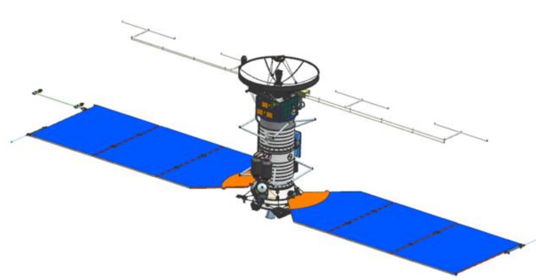

Figure 1: Europa Clipper Spacecraft Concept.

The HRS has been used in previous interplanetary missions, most notably the Mars Science Laboratory (MSL). The HRS architecture, shown in Figure 2, consists of a mechanically pumped fluid loop that harvests waste heat from electronic boxes (heat source) and delivers the heat to components requiring thermal energy for temperature maintenance (heat sink). The fluid then continues on to a radiator to reject surplus energy or bypasses the radiator to return to the pump; the percent flow control of the fluid bypassing the radiator is done passively by a temperature sensitive bypass valve. Implementation of the HRS architecture reduces electrical heater power demands by harvesting power already being used by other spacecraft and instrument operations. Additionally, the HRS leads to a more isothermal and thermally stable spacecraft.

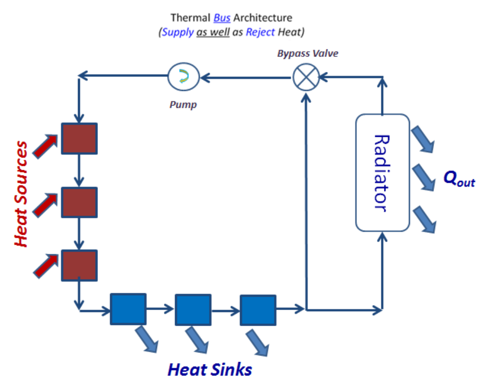

Figure 2: Heat Rejection and Harvesting System (HRS).

There are technical and non-technical risks associated with both the HRS thermal architecture as well as the passive one, which need to traded-off. Hardware fabrication and implementation can potentially be a lot more strenuous and complex for an HRS architecture than for a passive thermal architecture. In addition, tubing failure (e.g. due to micrometeoroid impact or other potential leak paths), can lead to deleterious impacts on the spacecraft's integrity. Finally, the HRS architecture has not been used in radiation environments as extreme as those found near Jupiter; but results from initial radiation risk mitigation studies are very promising. Hence, as the mission concept design and spacecraft system needs and resources evolve, it is important to weigh the benefits and potential risks of an HRS thermal architecture versus the passive one for the reference spacecraft and mission concept.

A study was commissioned by the project to trade the technical and non-technical aspects of employing an active thermal architecture and a passive thermal architecture for the reference Europa Clipper Spacecraft. This paper focuses on the technical aspects of the trade study – specifically on predicting the power demands and accompanying mass impacts for both architectures.

II. Spacecraft Configuration

Three modules make up the entirety of the Europa Clipper reference spacecraft (Figure 3). The Radio Frequency (RF) module houses the main components of the spacecraft telecommunication system: High Gain Antenna (HGA), Traveling Wave Tube Amplifiers (TWTAs) and waveguides. Under the

RF module sits the avionics module which along with the Stellar Reference Units (SRUs), Inertial Measurement Units (IMUs) and batteries, also houses the avionics vault where the majority of the spacecraft and instrument electronics will be located, (Figure 4 and Figure 5). The majority of the instruments are also expected to be located within the avionics module on a Europa Nadir pointing platform. The remainder of the spacecraft belongs to the propulsion module; fuel tanks, pressurant tanks, reaction wheel units, thermal radiators, the two solar array wings are located here.

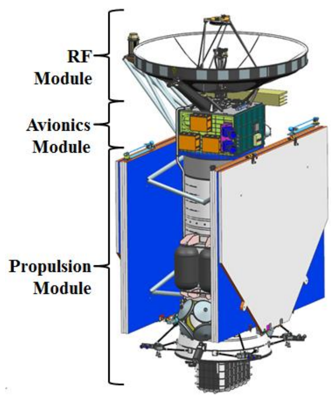

Figure 3: Three modules of the reference spacecraft.

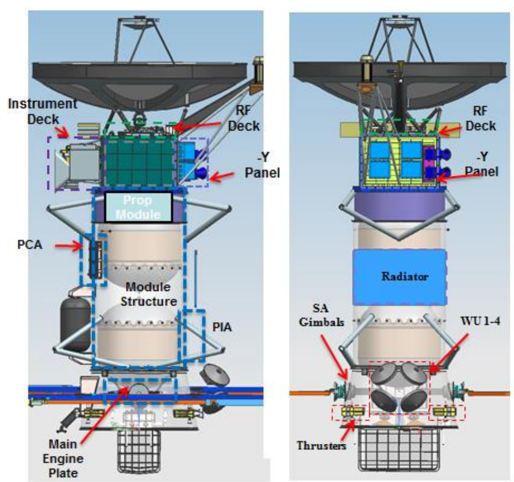

Figure 4: Major spacecraft Components.

Although most of the spacecraft power dissipation occurs in the vault, the majority of the spacecraft heat leak is through the propulsion module's MLI. The decoupling of the power source from the power starved components in the spacecraft mechanical configuration is made possible by the HRS active thermal control architecture.

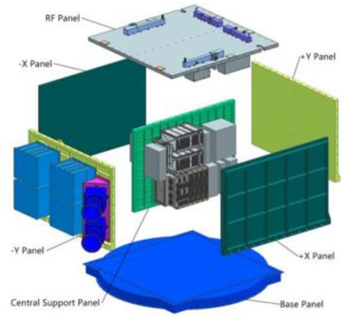

Figure 5: Vault Assembly Configuration.

III. Europa Clipper Active Thermal Architecture

The HRS active architecture uses a single phase mechanically pumped fluid loop routed across the spacecraft as shown in Figure 6. The working fluid (Trichlorofluoromethane / CFC-11) flows through 9.5 mm diameter pipes at 0.75 lpm. As the working fluid travels through the avionics and RF modules waste heat is harvested from the electronics and RF components. From there, the fluid makes its way to a replacement heater block where, when necessitated by various mission operational scenarios, additional heat is provided to the fluid to maintain the overall fluid loop temperature at the necessary level. The fluid loop is then routed to the propulsion module: the thermal sink for the fluid loop. Any excess heat is then radiated out to space before the fluid returns back to the vault.

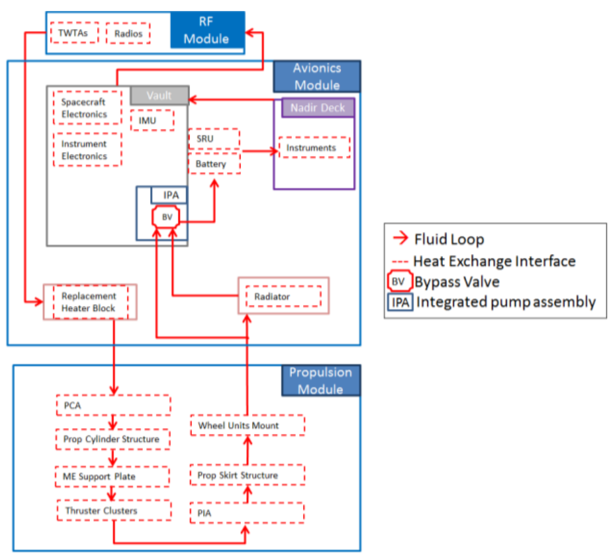

Figure 6: Fluid loop routing for Europa Clipper.

The fluid loop temperature is modulated using two separate components: the heater block and the bypass valve. When the fluid loop temperature is too high, the bypass valve diverts more fluid flow to the radiator to dissipate excess heat out to space. During cold conditions, the bypass valve closes the radiator flow and a replacement heater block provides supplemental heat when not enough waste heat is available to be harvested from the spacecraft components. In order to prevent the working fluid from freezing, the valve cannot be entirely closed; hence trickle flow to the radiator will act as a spacecraft heat leak that must be accounted for.

IV. Europa Clipper Passive Thermal Architecture

There are several alternatives to an active fluid-loop based thermal control architecture. However for the scope of this trade study the focus was to avoid the use of Radioisotope Heating Units (RHUs) and heat pipes; the former was a request by the management team of the pre-project study; the latter requires additional thermal analysis and design iteration that was considered too detailed for the quick turnaround time required for this trade study. Hence as an alternative to the active architecture, the study focused on a passive thermal architecture that uses localized louvers and radiators to maintain all electronics below their maximum allowable flight temperatures during hot conditions along with the use of electrical heaters to maintain all spacecraft components above their minimum allowable flight temperatures during cold conditions. This scheme does requires a change in the baseline vault configuration since most power dissipating components are nested inside the vault with no direct access to a space radiating surface as shown in Figure 7. These components must be removed from the center panel of the vault and relocated to the +X, -X, +Y,-Y faces of the vault; a configuration that is much more similar to the Juno spacecraft.

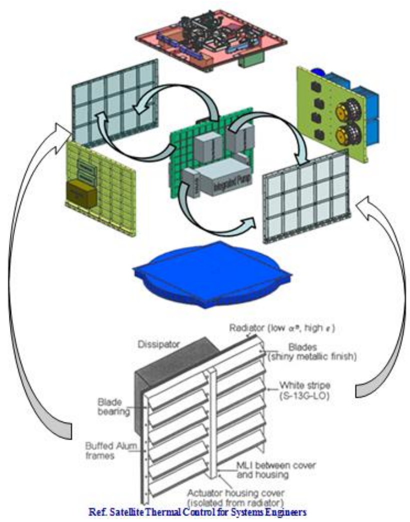

Figure 7: Vault Assembly reconfiguration for the passive thermal control architecture. All electronics mounted on the center panel must be relocated to side panels that would act as louver-controlled surfaces.

As much of the electronic heat dissipation occurs at the vault, the thermal coupling between the vault and upper region of the propulsion module will dictate just how much heat can be harvested from the electronics. A passive thermal design with a large thermal conduction path between the vault and propulsion module will require less additional heater power in order to maintain the propulsion components within their allowable temperatures. Additionally, a passive thermal architecture benefits from grouping components with similar temperature requirements together and isolating groups with different requirements from one another.

V. Trade Study Key Assumptions

The mission and spacecraft design are still in the concept phase. As a result several broad assumptions for key thermal analysis parameters have been made for the trade study.

Temperature Requirements Assumptions

The allowable flight temperature (AFT) assumptions for the major spacecraft components expected to be on the loop are shown in Table 1. The restrictive temperature limits are the batteries maximum allowable flight temperature of 35 ˚C and the propulsion tanks minimum allowable flight temperatures of 0 ˚C. It is assumed that the payload mounted onto the Nadir deck will be able to accommodate the temperature limits described in Table 1.

Table 1: Allowable Flight Temperatures (AFTs) for key spacecraft components.

Spacecraft Component	Minimum AFT [˚C]	Maximum AFT [˚C]
RF Deck	-35	50
Nadir Deck	0	50
Vault	-40	50
Batteries	-20	30
SRU	-20	45
IMU	-20	50
Prop Structure 1	0	50
Prop Structure 2	0	50
Prop Structure 3	0	50
Pressurant Tank	0	50
PIA	0	50
PCA	0	50
Thrusters	0	50
ME Plate	0	50
Wheel Units	0	65
SA Gimbal	-55	50

Additionally, the propulsion subsystem also requires a certain temperature gradient within certain components to be held at all times: the Pressurant system (tanks and PCA) must be held 2˚C above the temperature of the propellant system. This gradient helps minimize the risk of propellant condensation occurring during the mission.

MLI Assumptions

The prime heat loss path of the spacecraft is radiative heat loss through a large Multi-layer Insulation (MLI) enclosure. The amount of MLI area required for the spacecraft was estimated using the mechanical computer aided design (CAD) model assuming that a 5 cm micrometeoroid protection standoff will be used in the final implementation of the spacecraft. It was assumed that both architectures would implement the same MLI with no substantial difference in area (besides the vault region). Depending on the size of the MLI for a region, different worst case effective emittance (ε*) values were used in the thermal model. These values themselves were obtained from modeling guidelines developed for the Cassini spacecraft and provided in the Cassini Thermal Control Design Handbook [3].

Thermal Conduction between Components

Because the spacecraft design has baselined the HRS as the spacecraft thermal control system, much of the iteration focus between thermal and mechanical has been focused on accommodation of the fluid loop to the regions and components needing thermal control. That being the case, not as much iteration effort has been placed on determining the optimal thermal interface between different spacecraft regions and components. In fact, a benefit of an HRS thermal architecture is that the spacecraft mechanical design has a much more defined interface to the thermal control system via the HRS fluid loop as opposed to in a passive architecture where both the mechanical and thermal designs are deeply intertwined.

This poses a key caveat to the trade study: the spacecraft mechanical design has not considered a passive thermal architecture design into its development. The conductance estimates used for the spacecraft structure node network, Figure 8, was a very critical parameter in understanding the thermal behavior of the spacecraft system – especially for the passive architecture design. As such, the thermal interfaces between spacecraft components are not optimized for a passive thermal design which may negatively impact the performance of the passive architecture in the trade study.

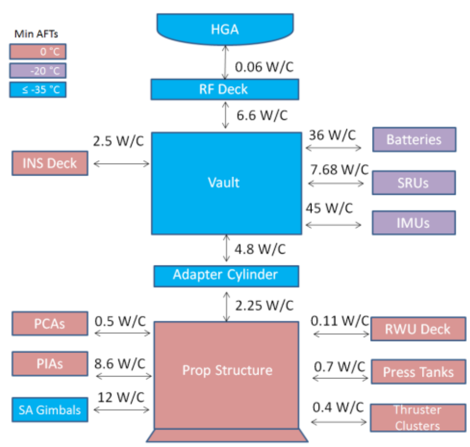

Figure 8: Total conductive couplings between main spacecraft components.

Environment and Power Modes

Two extreme environmental cases were defined to set bounds on the large variation in flight environments: a hot case environment identified as an inner cruise steady state at 0.6 A.U. with the spacecraft HGA nominally pointed to the sun with a zero degree off-sun angle; and a cold case environment defined as an outer cruise steady state at 5.6 A.U. with zero planetary thermal loads, again with the spacecraft HGA nominally pointed to the sun.

In addition, fifteen non-science nominal power scenarios for the mission concept had to be distilled into the most bounding cases, shown in Table 2. Both a steady state scenario and a transient scenario were identified for both outer cruise and inner cruise phases of the mission. The power conditioning unit (PCU) located in the vault acts like a 10% tax on the total solar array power generation. Hence the PCU is not in use when the spacecraft is under an Eclipse.

There are additional power scenarios specifically related to science instruments operating during Europa Flybys. Not surprisingly, these are also the most peak power demanding cases. Over forty flybys are planned with each flyby having a different instrument observation schedule. The schedules are sequenced such that each instrument only turns on below certain altitudes, however as not every flyby reaches the same minimum altitude, each flyby power scenario is noticeably different. In order to set bounds on the flyby cases, a worst case flyby scenario was generated that incorporates the longest expected operational time for each instrument, (Figure 9). Additionally, another short flyby scenario was generated to test for sensitivity of the system thermal power to electrical power demand.

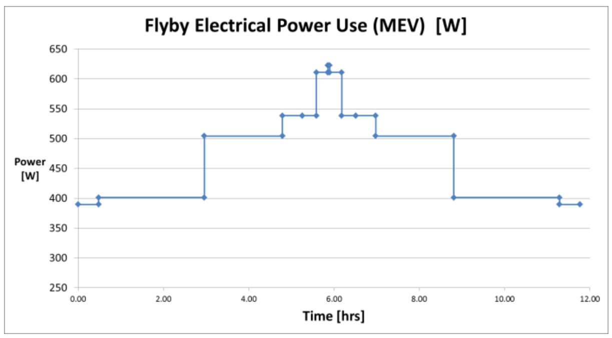

Figure 9: Flight system power demand during a representative worst case flyby (includes PCU demand). MEV = Maximum Expected Value

Table 2: Spacecraft electrical power modes, no thermal power demands shown. Notes: * IPA applies only in the active architecture configuration. **PCU is a tax on total solar array power generation and not active during eclipses.

Component	Inner Cruise		Outer Cruise	
	Steady State [W]	Transient [W]	Steady State [W]	Transient [W]
RF Deck	9	62	9	9
Vault	79	80	114	114
Instrument Deck	17	17	17	17
IPA*	14	14	14	14
SRU	4	4	4	4
IMU	36	36	36	36
PIA	26	26	6	6
PCA	1	1	1	1
ME Plate	0	0	0	0
Thrusters	0	0	0	0
SA Gimbal	13	13	13	13
Other-"Off-Loop"	9	57	37	37
PCU**	10%	10%	10%	0%
Total	**208**	**309**	**251**	**251**

VI. Thermal Analysis

Models

Two separate models were created using the ThermXL add-on package for Microsoft Excel: an active thermal architecture model and a passive thermal architecture one. Both models were able to run the two different environmental conditions and the four different power modes. Nodalization was done such that the larger components or regions of components with very similar thermal requirements were grouped. For example the vault, which houses all electronic boxes, was modeled as one single node while the propulsion module structure was divided into four nodes due to its large size and different components mounted onto it at noticeably different locations. Each component on the loop was also captured as a lumped node. Mass estimates were obtained from the Current Best Estimate (CBEs) captured in project's Mass Equipment List. The worst case hot and cold environmental conditions were stacked with appropriate worst case hot and cold parameters such as MLI e* and Radome solar absorptivity, (Table 3).

Table 3: Model parameters for Hot Cases and Cold Cases.

Parameter	Inner Cruise	Outer Cruise
A.U. Distance [A.U.]	0.6	5.5
Solar Flux [W/m^2]	3799	45
HGA Radome solar absorptivity	0.49	0.1
MLI > 1m^2, e*	0.01	0.02
MLI < 1m^2, e*	0.025	0.035
Louver e*	0.37	0.14

Other parameters (Table 4) such as MLI area and linear and radiation conductors were estimated using the mechanical CAD model for the reference spacecraft. The CAD model was also used to estimate the appropriate fluid loop routing for each component on the HRS thermal loop. The estimate for the fluid conductors was performed assuming CFC-11 fluid flows through 9.5 mm diameter pipes at 0.75 lpm when at 100% flow. Finally the HRS interface conductors, radiator fin effectiveness and louver form factor efficiencies were estimating using previous flight project analysis and experiments. The model heat balances were validated and a simplified eight-node excel model was used to validate results of overall heat losses to space.

Table 4: General Thermal Model assumptions.

Active Model Parameters		Passive Model Parameters	
S/C MLI Area [m^2]	33	S/C MLI Area [m^2]	32
Radiator Area [m^2]	0.32	Louver Area [m^2]	1
Radiator Fin eff	0.77	Radiator Fin eff	0.77
Bypass Set Points [˚C]	0-15	Louver packing eff	0.8
Fluid Conductors [W/C]	16.6	Louver Set Points [˚C]	2-20
HRS Interface G [W/(m-C)]	2.2		

Thermal Analysis Methodology

The steady state hot and transient hot case were used to estimate the size of the radiator or louver, depending on the architecture, such that low to no temperature margin was kept at the end of the hot case transient (bounding case). Once the radiator was sized for the active configuration and the louver set points were determined for the passive configuration, the steady state cold case was investigated to determine the necessary supplemental heater power needed, either in the replacement heater block for the active architecture, or as localized heaters for the passive architecture. Again, the heater power was sized such that low to no temperature margin was kept. These cold case steady state results were set as the initial conditions for the transient eclipse and flyby model runs.

VII. Results

Power demand

The heat balance results for the steady state cold case are shown in Figure 10. For the active configuration, 123 W are harvested from the avionics module and transported to the propulsion module, with radiator loss occurring due to trickle flow. Without the HRS fluid loop, the avionics module loses the majority of its power dissipation through the surrounding MLI. However the propulsion module still requires about 200 W to be kept at its minimum AFTs. Hence for the steady state cold case, a total of 67 W of additional thermal subsystem power is needed for the passive architecture over the active architecture. Because of the PCU dissipation percent penalty, this accounts for a flight system that requires 75 W of additional power for a passive thermal architecture.

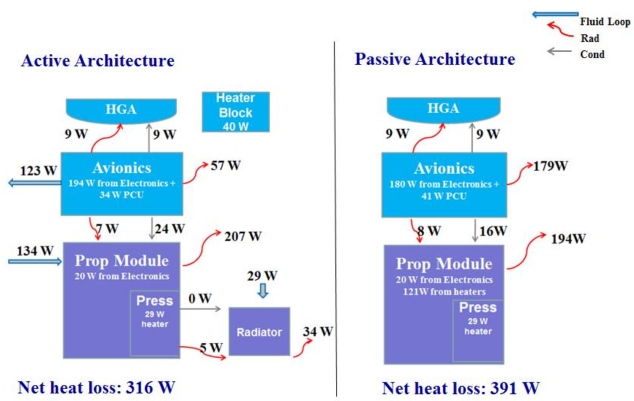

Figure 10: Steady State Cold Case Heat Balance Maps of both thermal architectures.

The transient thermal subsystem power demand estimates are listed in Table 5. For the eclipse case, thermal power demand decreases for both the active and passive architectures as you are able to "ride-out" the transient. However, because PCU dissipation is inactive during the eclipse, the active architecture cannot reduce its heater power demand as it has lost the heat it harvests from the PCU. The passive architecture on the other hand, does not rely on PCU dissipation to heat the propulsion module and hence electrical heater power demand can decrease.

Table 5: Comparison of Thermal subsystem power demand for both architectures.

Thermal Subsystem Power Demand		
Power Scenaro	**Active Arch. [W]**	**Passive Arch. [W]**
Steady State Cold	83	150
Transient Cold	83	129
Peak Flyby	5	157

The flyby scenario shows the most extreme difference between the active architecture and passive architecture especially at the peak power phases of the flyby. The active architecture is able to reduce its electrical heater power demand to almost zero as it can harvest power from instrument electronics that start operating during the flyby. For the passive architecture on the other hand, the additional power dissipation from the instrument electronics only result in localized heating near the vault. This leads to the louvers opening up more to allow the additional power to dissipate out to space. The localized heating also prohibited the option to “ride out” the flyby transient. As the upper region of the fuel tank heated, the PCA and Pressurant tanks also needed to be heated in order to keep them 2˚C above the propulsion tanks. This in the end created a heater power demand increase during the flyby transient as opposed to an expected decrease. Note that this fuel tank localized heating also occurred in the active configuration; local electrical heaters at the PCA and Pressurant tanks had to be used in the flyby case to meet the 2˚C gradient requirement. However, the fuel tank localized heating was not as pronounced due to the HRS maintaining the spacecraft more isothermal than the passive thermal architecture.

Mass Impacts

The subsystems with mass impacts due to the decision of an active or passive thermal architecture are thermal, mechanical, and power. Overall, a passive thermal architecture was estimated to require an additional 110-150 kg (CBE) versus an active thermal architecture.

The mass estimates for the thermal hardware necessary for each architecture was estimating per the knowledge of previous missions with similar hardware; specifically the active thermal architecture hardware of MSL and the passive thermal architecture hardware of Juno were used as the primary references for thermal hardware mass estimates. The big line item for mass impact is the mechanically pumped fluid loop. Per these reference missions, the passive thermal subsystem mass is estimated to be 39 kg (CBE) less than an active architecture.

The power subsystem’s mass impact came from having to accommodate the additional power demand for the passive architecture and maintain the same power and energy margins. The 67 W required in the steady state required an additional 88 kg (CBE) for the Solar Array size increase. The power subsystem also requires an additional battery, 22 kg (CBE), to accommodate the transient power demand for the passive thermal architecture. Hence for the power subsystem, the passive thermal architecture is estimated to require a 110 kg (CBE) more mass versus the active architecture.

Due to the quick turnaround time of the trade study, only a zeroth order estimate could be completed for the mechanical subsystem's mass impact. Mechanical accommodation of a passive thermal architecture would require changes in the vault configuration; relocating the electronics boxes from the center panel of the vault to the outer panels requires an increase in mounting area for the outer panels. Additional mechanical accommodations may also be deemed necessary (such as doublers on mounting surfaces). For the mechanical subsystem, the passive thermal architecture is estimated to require 40-80 kg (CBE) of additional mass versus the active thermal architecture.

Hence although the thermal subsystem's mass is less for a passive architecture, the additional mass needed for mechanical and power to accommodate the architecture is substantially more. An overall flight system mass upper of 110kg-150 kg (CBE) is expected for a passive thermal architecture.

VIII. Conclusion:

An active thermal control active architecture using an HRS fluid loop was traded with a passive thermal architecture for the Europa Clipper reference spacecraft. The thermal subsystem power demand estimate is lower for the active architecture than the passive architecture: 67W, 46W, and 113W less for the cold case steady state, transient eclipse cold case, and the peak flyby case. The difference in power demands is due to the HRS architecture able to harvest waste heat from the electronics; during the flyby case the instrument electronics supply additional waste heat that substantially reduces electrical heater power demand for the active architecture. Additionally, substantial localized heating in the fuel tank occurs in the passive architecture. This requires supplemental heater power to maintain the PCA and pressurant tanks 2˚C above the fuel tank.

The results of the power analysis were used to determine solar array and battery mass impacts. To maintain the same mass margins, the power subsystem was found to require an additional 110 kg (CBE) of mass for the passive thermal architecture versus the active architecture.

Acknowledgments

The development and analysis described in this paper was carried out at the Jet Propulsion Laboratory, California Institute of Technology, under a contract with the National Aeronautics and Space Administration. The authors express their thanks to Daniel Lok at Applied Science Laboratory, Bruce Williams at Applied Physics Laboratory, Matthew Spaulding at JPL, Alexander Eremenko at JPL, Antonio Ulloa-Severino at JPL, Roxanne Arellano, and the rest of the Europa Clipper Mission Concept team.

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